TITLE

Novel mission concepts for polar coverage: an overview of recent developments and possible future applications

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ABSTRACT

The paper provides a survey of novel mission concepts for continuous, hemispheric polar observation and direct-link polar telecommunications. It is well known that these services cannot be provided by traditional platforms: geostationary satellites do not cover high-latitude regions, while low- and medium-orbit Sun-synchronous spacecraft only cover a narrow swath of the Earth at each passage. Concepts that are proposed in the literature are described, including the pole-sitter concept (in which a spacecraft is stationary above the pole), spacecraft in artificial equilibrium points in the Sun-Earth system and non-Keplerian polar Molniya orbits. Additionally, novel displaced eight-shaped orbits at Lagrangian points are presented. For many of these concepts, a continuous acceleration is required and propulsion systems include solar electric propulsion, solar sail and a hybridisation of the two. Advantages and drawbacks of each mission concept are assessed, and a comparison in terms of high-latitude coverage and distance, spacecraft mass, payload and lifetime is presented. Finally, the paper will describe a number of potential applications enabled by these concepts, focusing on polar Earth observation and telecommunications.

Keywords: Continuous polar coverage, high-latitude observation, Earth observation, Polar meteorology, Mission design

MANUSCRIPT

1. INTRODUCTION

Spacecraft in geostationary orbit (GEO) have demonstrated the immense possibilities offered by the continuous coverage of a particular region. GEO platforms are nowadays the most used satellites for broadband telecommunications and weather forecasting. Unfortunately, GEO platforms can only provide their services in the equatorial and temperate zones, where elevation angles are sufficiently high. At higher latitudes, similar services are provided at present only by satellites in highly-inclined or polar, low or medium orbits. These orbits, such as Sun-synchronous orbits, allow the spacecraft to image only a narrow swath at each passage, relying on multiple passages for full coverage. For example, Landsat 7 (altitude of 705 km at 98.2°) completes just over 14 orbits per day, covering the entire Earth between 81 degrees north and south latitude every 16 days[1]. At each passage, the spacecraft can observe only a narrow swath of surface, and therefore multiple passages are required for obtaining a full polar image. Consequently, the temporal coverage of the entire polar region can be poor, as different areas are imaged at different times, hence missing the opportunity to have a simultaneous and continuous real-time complete view of the pole. At present, these images are post-processed to make a "composite" image, which can be used, for example, for weather forecasting and wind vector prediction. However, the data that can be extracted is neither complete nor accurate [1].

To overcome these issues, it is desirable to have a spacecraft with a continuous view of the poles, or even better, one that is constantly above one of the poles, stationary with respect to the Earth, in the same way as a GEO spacecraft is stationary above one point on the equator. This spacecraft is known in literature as "pole-sitter". The first study of this concept appears to have been made by Driver [2] in 1980, although the author claims that the original idea belongs to the mathematician and writer Kurd Lasswitz. In the following years, the idea was then extended by other authors, introducing new mission concepts, either by releasing the strict constraint of the positioning of the spacecraft, or by introducing new forms of propulsion (solar sailing [3], or hybrid propulsion [4]) or a combination of both [5, 6], to increase the mission lifetime, decrease the launch mass, or increase the visibility conditions of the polar regions, in terms of coverage and resolution.

The paper is organised as follows. The first section is a brief overview of low-thrust propulsion technologies. The second and most extended section of the paper attempts to provide a comprehensive overview of mission concepts for high latitude and polar Earth observation and telecommunications. It will not only focus on the pole-sitters, in the strict sense, but also on other concepts, in particular to those exploiting non-Keplerian orbits and the circular restricted three-body problem (CR3BP). The paper will critically describe the concepts presented starting from 1980 onwards, comparing feasibility and performance in terms of polar coverage performance, thrust required, and spacecraft mass for a given payload size and lifetime. Molniya orbits [7] will also be included, for two reasons: the first is that they have been historically used for high-latitude telecommunications; second is that they have recently been extended with a novel concept [8]. The overview also includes some recent results obtained concerning natural and solar sail-displaced orbits in the CR3BP. Finally, the paper briefly introduces some possible mission applications

[1] Landsat 7 Handbook, http://landsathandbook.gsfc.nasa.gov/ [Cited 12/09/2011]

that can be enabled by continuous polar observation, including observation, polar meteorology and telecommunications.

2. PROPULSION SYSTEM TECHNOLOGY

Many of the concepts presented in the following require a continuous acceleration, to keep the spacecraft at a stationary point (e.g. constantly above the pole) or along a non-Keplerian orbit, counterbalancing other forces. We provide here a brief overview of the two propulsion systems that will be exploited: solar electric propulsion (SEP) and solar sailing.

Solar electric propulsion is a mature technology that provides a spacecraft with a relatively low thrust (of the order of a fraction of a Newton per thruster [9]), by accelerating propellant to very high speed. Despite the relatively high specific impulse of modern thrusters [9] (of the order of 3000-8000 s), the thrusting time and hence the mission duration is always limited by the mass of propellant on-board: the lifetime L is directly related to the propellant mass fraction m_{prop}/m_0, the required acceleration a (assumed constant), and the specific impulse I_{sp} through the scaling law:

$$L = -\ln\left(1 - \frac{m_{prop}}{m_0}\right)\frac{I_{sp} g_0}{a} \tag{1}$$

where $g_0 = 9.81\ \mathrm{m/s^2}$.

In the attempt to increase mission lifetime, many concepts propose to adopt a solar sail: this device can provide a continuous thrust without the use of any propellant mass. The idea, dated back to the beginning of 20th century and investigated in detail [10] since then, is in principle simple: gaining momentum by reflecting the photons from the Sun.

In the case of a flat, perfect sail, the acceleration is directed normal to the sail and away from the Sun, and its magnitude is [10]:

$$a_s = \beta \frac{\mu}{r^2}\cos^2\alpha$$

where μ is the gravitational parameter of the Sun, r the Sun-spacecraft distance, and β is the lightness number, a function of the sail loading $\sigma = m/A$ of the spacecraft (mass over sail area):

$$\beta = \sigma^*/\sigma$$

and $\sigma^* = 1.53\ \mathrm{g/m^2}$ is a constant for the Sun. The cone angle of the sail $\alpha \in [0, 90°]$ measures the angle between the sail normal and the Sun direction. When the sail is flat towards the Sun at 1 AU ($\alpha = 0°$), then the acceleration produced by the sail is known as the characteristic acceleration (a_0). The conversion between β, σ and a_0 at 1 AU is represented in Fig. 1. Any of these three parameters are an indicator of the technology needed for the spacecraft: the larger the lightness is, the lower the sail loading is. This is achieved either using a larger sail area, or by reducing the system or sail mass. Values of β up to 0.05 can be assumed for a near-term system. Recently flown solar sail demonstrators, however, had considerably lower lightness numbers: JAXA's IKAROS [11] has a 20-m-diagonal square sail and weighs 350 kg ($\beta = 0.001$), while NASA's NanoSail-D2 [12] is 4 kg for 10 m^2 ($\beta = 0.003$).

Note that in the case of a hybrid SEP/sail spacecraft, the mass of the spacecraft varies during its lifetime due to propellant consumption, and so does β. For this type of spacecraft, the reference value will be the one at the beginning of life, when it is smallest.

The areal density of the sail assembly σ_s, instead, defines a technological parameter for the sail assembly. It is expected that technological developments [13] should enable sails of 10 g/m^2 in the near future. Ultra-thin (around 2 μm of thickness) sails are expected in the mid- to far-term timeframe [14] and can lead, for large sails, to loadings on the order of 5 g/m^2.

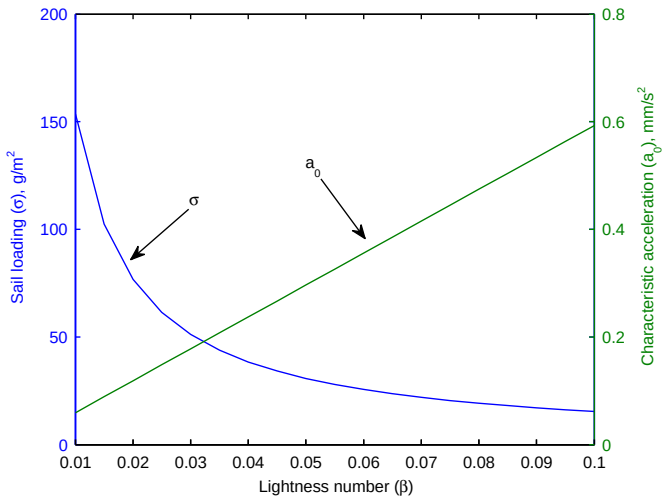

Fig. 1. Conversion between lightness number (β), sail loading (σ) and characteristic acceleration (a_0) at 1 AU.

3. CONTINUOUS POLAR OBSERVATION CONCEPTS

In this section, we will review and compare the concepts that have been proposed in the literature for continuous polar observation.

Concepts proposed in the CR3BP use the gravitation of both the Earth and the Sun, and exploit the complex dynamics of the system to find favourable positions or orbits, see Fig. 2. It is worth underlining that whenever we refer to the CR3BP, the reference system is synodic, i.e. co-rotating with the Sun-Earth line. This is the reason why, in Fig. 2, the polar axis of the Earth, whose direction is considered fixed in time, rotates once every year, describing a conical surface. In this and following trajectory plots, the x axis is the Sun-Earth line, pointing towards the Earth, and the z axis is aligned with the angular velocity of the two bodies.

Polar coverage will be assessed in terms of the latitude of the spacecraft throughout the year, or the period of the orbit, and in terms of the minimum latitude (in absolute value) at which a continuous view of the pole is available as the Earth rotates around its polar axis (see Fig. 3). This latitude, defined as Λ, is computed taking into account a minimum elevation angle of the spacecraft, ε. In this paper, ε is set to the reference value of 27°, which is the same as that of a GEO spacecraft at latitude of 55°.

It is desirable to have a low (in modulus) value of Λ, as this would mean continuous coverage not only of the polar caps, but also lower latitude regions around the pole. Λ might not be defined, if the latitude φ of the spacecraft is so

low (with respect to its distance) that no continuous coverage is possible at any latitude. Finally, if the angle Λ varies with time during the mission, then its maximum (in absolute value) shall be considered, as it represents the worst case scenario.

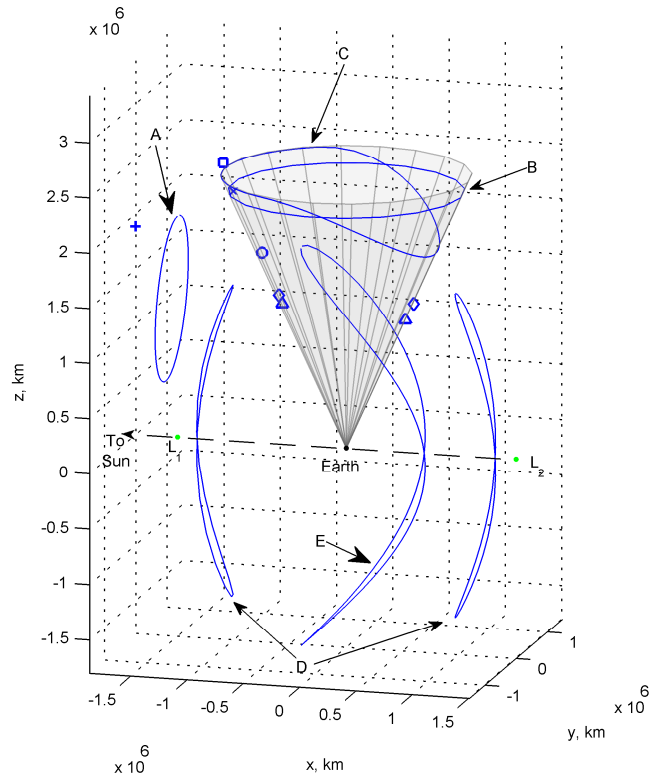

Fig. 2. Proposed concepts for polar observation in the CR3BP. +: Sail [3], β = 0.05. ×: Sail [3], β = 0.1. ○: SEP [6], a = 0.18 mm/s^2. ◊: Hybrid [6], β = 0.03, a = 0.18 mm/s^2. □: Hybrid [6], β = 0.03, a = 0.086 mm/s^2. △: Hybrid [6], β = 0.05, a = 0.18 mm/s^2. A: Sail [5], β = 0.051. B: SEP [2], a = 0.16 mm/s^2 (average). C: Hybrid, β = 0.05, a = 0.11 mm/s^2 (average). D: Natural [15]. E: Sail, β = 0.02.

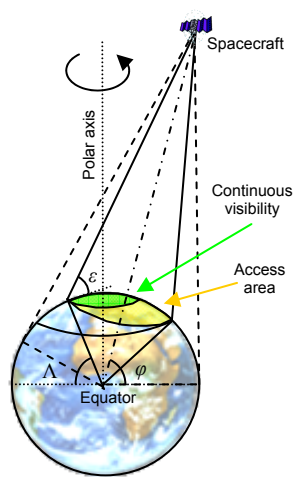

Fig. 3. Geometry of the spacecraft coverage. The yellow area is the surface covered with minimum elevation angle of ε. The green area is the surface that is continuously visible from the spacecraft with the same minimum elevation angle: therefore Λ is the minimum latitude of continuous coverage.

3.1. SEP pole-sitter

The idea of a stationary spacecraft above a pole of the Earth first appeared in 1980 by Driver [2]. In that study, the author simply assumed a spacecraft stationary on the polar axis of the Earth, and systematically computed the acceleration required to counterbalance the dominant perturbations, including the Earth's gravity, as a function of the distance from the Earth and the time of the year. A SEP system was assumed, which can provide continuous thrust while propellant is available.

It was shown that the minimum magnitude of acceleration is for distances ranging from 2.29 to 2.74 million km, depending on the time of the year. Within this range, the acceleration needed varies from 0.155 to 0.166 mm/s^2.

If the distance is kept constant, the minimum propellant mass fraction per year is obtained at a distance of about 2.5 million km, as illustrated in case B of Fig. 2. As the distance shortens, the acceleration needed becomes quickly unrealistic.

Due to the large distances of the spacecraft to the Earth, the full hemispheric visibility is limited only by the minimum elevation angle, ε. In particular, since the distance of the spacecraft is much larger than the radius of the Earth, the minimum latitude at which the hemispheric image extends is roughly the same value as the minimum elevation angle (see Fig. 3, in which for a pole-sitter the latitude of the spacecraft is always 90°), i.e. $\Lambda \cong \varepsilon$. Therefore we can say that a pole-sitter spacecraft has a full hemispheric view of all latitudes above 27°, at any time of the mission.

At 2.1 million km (or about 5 lunar distances), where the acceleration is close to its minimum, 24 kW is sufficient to power 6 SEP thrusters with $I_{sp} = 3000$ s for 2.3 years. However, due to the high power needed by the SEP system to generate the necessary acceleration, the mass of the whole system is relatively large: a preliminary mass budget states that the spacecraft would weigh about 4100 kg at the beginning of the mission at the pole, of which 3100 kg is the propulsion system (propellant, thruster, solar panels) and the remaining 1000 kg is the net mass of the spacecraft (payload and remaining subsystems).

Driver also considers different scenarios in which the distance of the spacecraft is lower. For example, at 1.1 million km (or about 3 lunar distances), where the acceleration is 0.35 mm/s^2, a spacecraft with $I_{sp} = 5000$ s can last for about 1 year; the initial mass is 5400 kg, again with 1000 kg of mass net of the propulsion system and propellant. The power required is 92 kW for 10 thrusters.

It is worth underlining that the mass budget for an SEP spacecraft depends on a number of assumptions, and therefore it can change consistently depending on the way it is computed, and the technology parameters assumed. For sake of completeness and comparison, in Fig. 4 we provide a graphical mass budget considering that the spacecraft delivers a constant acceleration through the whole lifetime. The figures are plotted using analysis and values found in Ceriotti et al. [16], and they match to what was used in the works described in the following Sections 3.4 and 3.5.

There are two major drawbacks in this concept: the first, and most important, is the considerable distance at which the spacecraft shall stay, of the order of 1-3 million km. The second is the relatively high level of acceleration needed throughout the whole mission, which limits the mission lifetime to a few years.

There is another issue mentioned in the paper, which is related to the thruster pointing direction: in fact, twice a year, the thrust vector is directly exactly away from the Earth, and therefore the thrust nozzle is towards the Earth: in this configuration, the exhaust plume may interfere with the observation/telecommunication payloads, which are also pointed in the same direction.

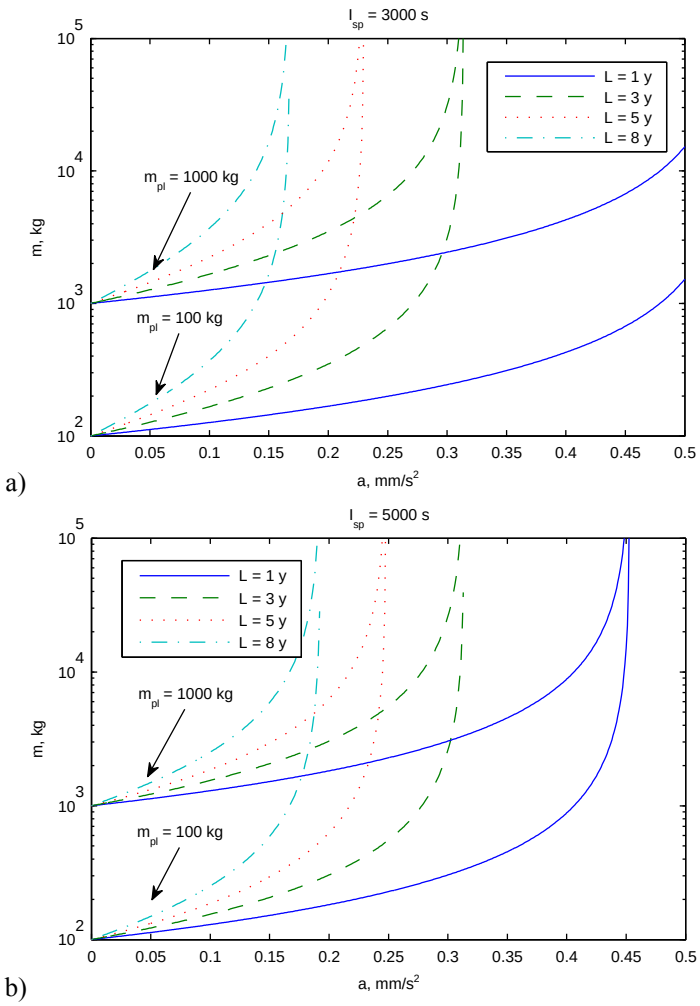

Fig. 4. Mass budget of an SEP spacecraft providing constant acceleration, for different payload capacity and lifetime. Specific impulse of 3000 s (a) and 5000 s (b).

3.2. Statite

Robert Forward, in 1991 [3], published a spacecraft concept for polar communications, based on the surfaces of equilibrium of a solar sail in the CR3BP described by McInnes et al. [17]. The concept uses a solar sail to displace the natural equilibria of the CR3BP above (or below) the ecliptic.

Fig. 5 (a) illustrates the possible positions with contour lines, each one of which refers to a lightness number. As can be seen, realistic lightness numbers allow the placement of the spacecraft on the L_1 side (day side) at relevant vertical displacements, but not on the L_2 side (night side). Moreover, due to the limitations in the attitude of the solar sail, there is a "forbidden region" in which sails cannot be stationary. These curves can be slightly modified if a photon thruster is used, i.e. a two-mirror device. However the complexity of the system is considerably increased. It is also to be underlined that an ideal sail is considered in this work: the effect is an optimistic calculation of Fig. 5 (a). If a non-ideal sail is employed (i.e. photon non-specular reflection and absorption are considered), then the "forbidden region" expands, and contour lines contract.

The latitude φ and the angle Λ are represented in the dashed line in Fig. 6, for the spacecraft in the highest possible displacement achievable with $\beta = 0.05$ (see + in Fig. 2). Note that around winter continuous observation of the North Pole is not possible with $\varepsilon = 27°$. The southern case is symmetrical. Forward also proposes different modes of operation, in which the spacecraft is not stationary in the rotating frame, but moves slowly, for example to maintain a fixed latitude with respect to the Earth.

Being a pure sail spacecraft, the lifetime is not limited by the propellant onboard, and therefore arbitrarily long missions could be envisaged. It is however to take into consideration that the artificial equilibria are unstable, and therefore is likely that some conventional thrusters may be required for station-keeping.

Despite that Forward does not present a mass budget, it can be easily computed for a solar sail spacecraft, by splitting its total mass m into the sail mass m_s and payload mass m_{pl} (the rest of the spacecraft), and then considering that $m_s = \sigma_s A = \sigma_s \beta m / \sigma^*$. Fig. 7 presents the spacecraft mass as a function of the lightness number, which is fixed once the position of the spacecraft is established through the curves in Fig. 5. The other two variables define the payload that has to be carried onboard and the technology level of the sail.

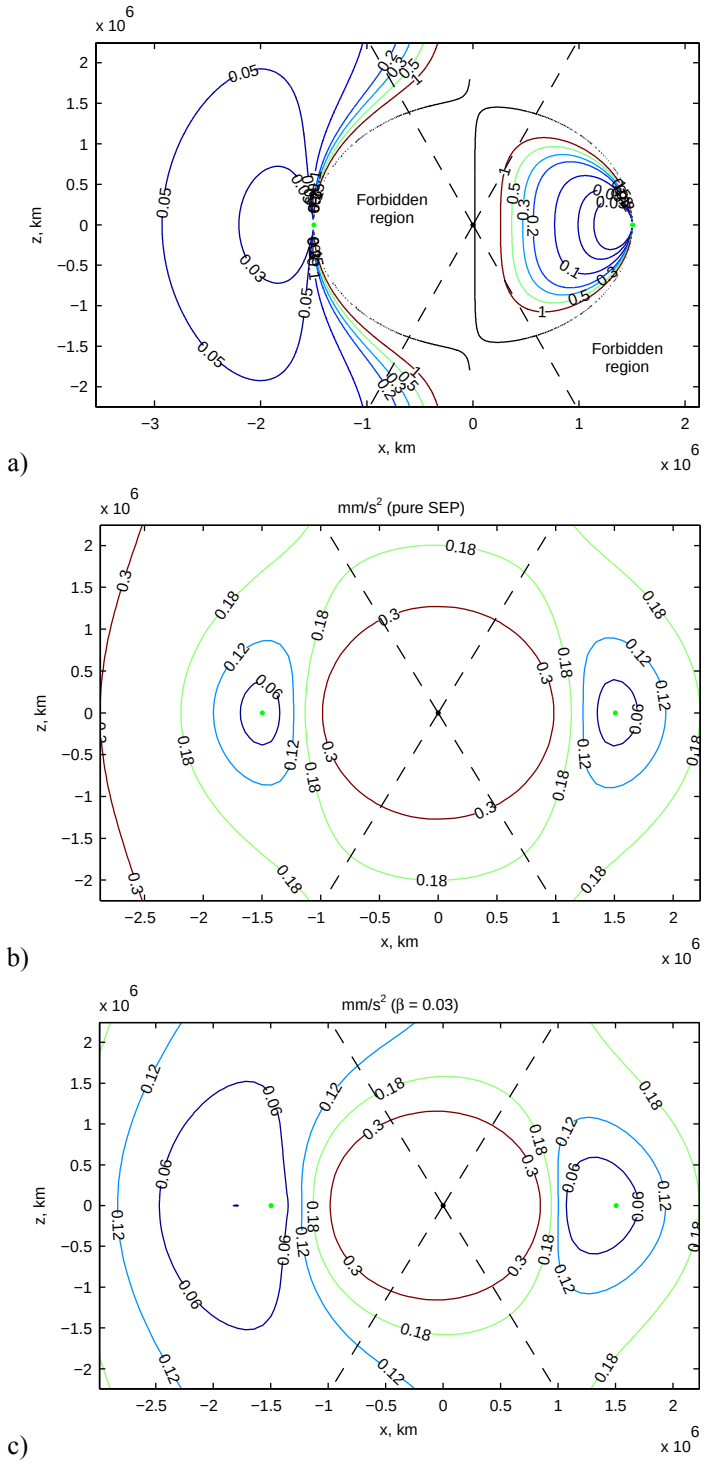

Fig. 5. Possible positioning of a static spacecraft in the CR3BP. a) Lines of constant lightness number of a pure sail [3]. b) Lines of constant acceleration (in mm/s^2) for a pure SEP. c) Lines of constant SEP acceleration (in mm/s^2) for a hybrid sail/SEP spacecraft [6] with $\beta = 0.03$ and non-ideal sail.

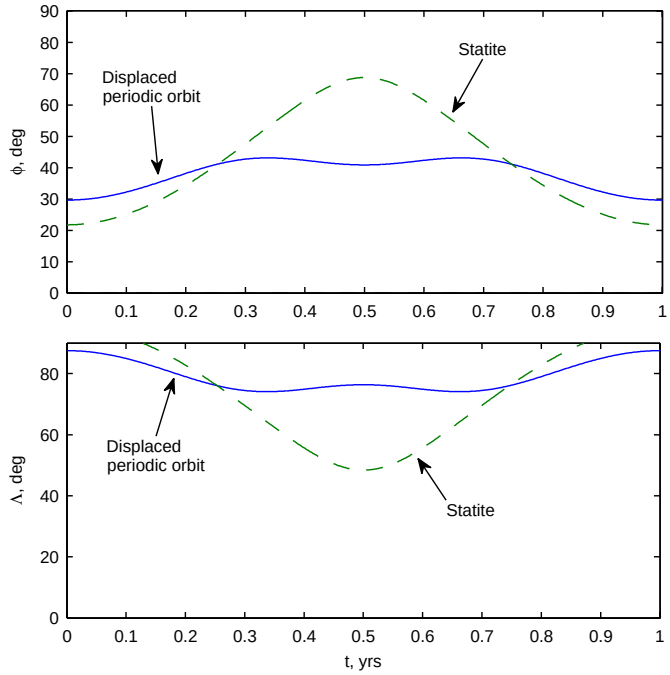

Fig. 6. Spacecraft latitude (φ) and minimum continuous visibility latitude (Λ), for the Statite spacecraft (dashed line, + in Fig. 2) and the periodic orbit around displaced equilibrium (solid line, A in Fig. 2).

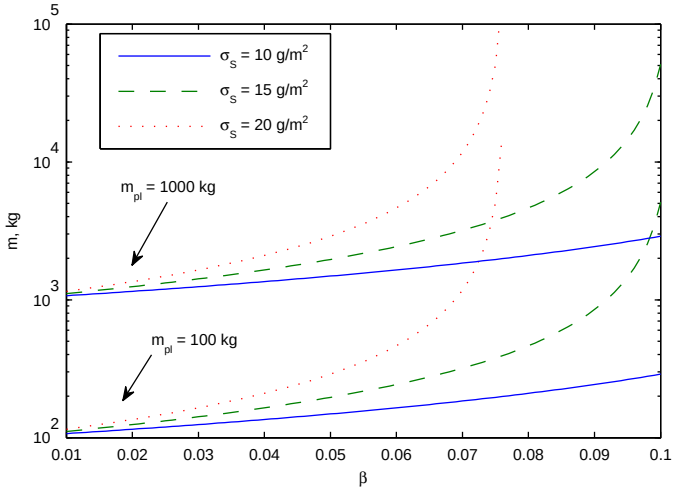

Fig. 7. Mass budget of a solar sail spacecraft: total mass *m* as a function of the lightness number *β*, for different sail areal densities (σ_s) and payload masses (m_{pl}).

3.3. <u>Periodic orbits around displaced equilibria</u>

This approach is an extension of the static concept by Forward. Since a solar sail creates artificial, displaced equilibria in the CR3BP, then periodic orbits can be designed around these points, in the same way Halo orbits exist around natural equilibria. In particular, Waters and McInnes [5] chose an artificial point above L_1 and found families of orbits with different amplitude and period: the one-year-period orbit ($\beta = 0.05174$) is particularly interesting,

because it allows the sail to move synchronously with the Earth's polar axis tilting, counterbalancing its movement, and hence reducing the variation in latitude of the spacecraft throughout the year (see A in Fig. 2).
On this orbit, the latitude of the spacecraft ranges from a minimum of 29° in winter to a maximum of about 43° in spring and autumn; refer to the solid line in Fig. 6. If 27° of minimum elevation angle is considered, then the minimum latitude of full coverage goes up to 87.5° in winter, being barely sufficient to cover the pole for the considered realistic lightness number. However, it represents a substantial improvement with respect to the performances that can be obtained for approximately the same lightness number, but in the static case proposed by Forward (see green line in Fig. 6).
As for the Statite spacecraft, a symmetric orbit can be employed for the South Pole (but not on the L_2 side), and the lifetime is potentially unlimited.
For an estimation of the mass of this spacecraft, we can refer again to Fig. 7.

3.4. Artificial equilibria for hybrid propulsion

In 2008, Baig and McInnes [6] developed an extension of the Statite concept by Forward, introducing hybrid solar sail and solar electric propulsion in the same spacecraft. The idea of this approach is that the two propulsion system can compensate each other, allowing both a better placement of the spacecraft with respect to the pure sail case, and propellant saving with respect to the pure SEP spacecraft.
As seen from Fig. 2 (a), a pure sail does not allow the positioning of the spacecraft in the "forbidden regions". Furthermore, large lightness numbers are needed to reach the 23.5° line of the Earth's polar axis. However, if an SEP spacecraft is considered (Fig. 5 (b)), then an acceleration of about 0.18 mm/s^2 will suffice to position the spacecraft at about 1.5 million km from the Earth either on the winter or the summer line. This is due to the fact that the SEP thruster can provide the acceleration in an arbitrary direction, without the limitations of a solar sail. If a sail (lightness 0.03) is added to this spacecraft, then the same position on the summer line can be maintained with an SEP acceleration of 0.12 mm/s^2, or alternatively an SEP acceleration of 0.18 mm/s^2 allows positioning closer on both winter and summer lines (Fig. 5 (c)).
The authors propose a mission in which the spacecraft is positioned at 2.7x10^6 km on the summer line (☐ in Fig. 2). Fixing a mission time of 5 years and a small payload of 100 kg, it is found that a pure sail with assembly areal density of 10 g/m^2 requires a spacecraft mass of 460 kg; a pure SEP spacecraft (Statite-like) with specific impulse of 3200 s requires 621 kg of total initial mass; finally, a hybrid spacecraft combining propulsion systems with the same characteristics requires only 288 kg of initial mass. However, the pure sail spacecraft mass does not depend on the mission duration, i.e. the length of the mission can be extended arbitrarily without substantial change in the mass budget: the same does not hold for the other two platforms. Note also that this work took into account also the non-perfect reflectivity of the solar sail.
The latitude of the spacecraft throughout the year and the minimum latitude for full visibility with a minimum elevation angle of 27° are represented in Fig. 8. Each curve corresponds to spacecraft positioned on the day-side and night-side line respectively, on the North Pole.

With reference to Fig. 8, if two spacecraft are employed, continuous visibility up to about 60° of latitude is possible. With only one spacecraft, the minimum latitude increases to 73°. Two symmetrically placed spacecraft are necessary for the South Pole.

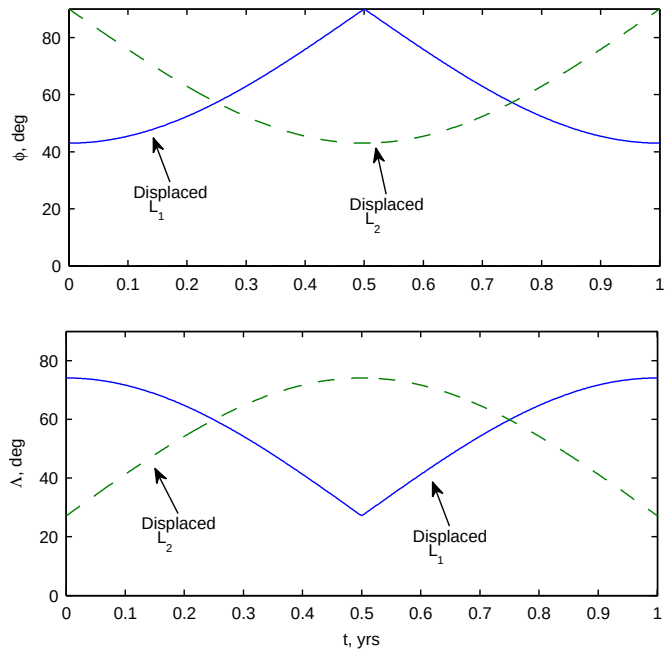

Fig. 8. Spacecraft latitude (φ) and minimum continuous visibility latitude Λ, for two spacecraft, above L_1 (solid line) and L_2 (dashed) respectively.

3.5. <u>Optimal hybrid pole-sitter</u>

This concept [4] combines the advantages of the hybrid propulsion, which allows a saving of propellant mass fraction with respect to the SEP-only case, and the true pole-sitter, which allows a full hemispheric view. Furthermore, by designing an optimal orbit that relates the distance of the spacecraft with the time of the year, it is possible to save further propellant. Optimal orbits depend on the initial mass of the spacecraft, as well as the sail lightness number. However, in general, all of them become closer to the Earth in winter and further from the Earth in summer, as shown in the orbit identified with C in Fig. 2. For sake of comparison with the hybrid equilibrium spacecraft of Section 3.4, and to prevent the spacecraft going too far from the Earth, it was decided to find optimal orbits constraining their maximum distance from the Earth to 2.7 million km.

A detailed mass budget for this concept [16] shows that a payload of 100 kg can be kept in a pole-sitter position for 5 years with a 81 mN SEP spacecraft of initial mass 465 kg, or with a hybrid spacecraft weighing 408 kg with 59 mN SEP thrust and a solar sail of $\beta = 0.035$ (but with a rather light assembly of 5 g/m^2). If instead a mission of 8 years is considered for the same payload mass, then the SEP spacecraft would weigh 3773 kg, as opposed to the hybrid propulsion spacecraft of 2871 kg and 1153 kg, for $\beta = 0.025$ (7.5 g/m^2) and $\beta = 0.035$ (5 g/m^2) respectively. The SEP thrust level is 660 mN for the SEP-only spacecraft, but lowers to 435 mN and 167 mN for the two hybrid configurations.

As for the other pole-sitter presented in Section 3.1, the hemispheric view is constantly available down to a latitude that is equal to the minimum elevation angle, i.e. 27°.

The hybrid propulsion applied to the pole-sitter, in addition to enabling longer missions or lighter spacecraft with respect to the pure SEP, has a side advantage, which becomes clear considering the direction of the resulting thrust that is needed to maintain the pole-sitter position; in the range of distances under consideration, this is mainly directed away from the Earth. As Driver [2] points out, this is an issue in some seasons of the year, when the SEP engine nozzle is directed towards the Earth, possibly interfering with telecommunication antennas and imaging payload. However, when a solar sail is added, the SEP thrust is tilted, leaving the direction towards the Earth free. Moreover, the tilting of the thruster nozzle with respect to the sail only changes 2° throughout the year, avoiding the use of a gimbal capable of large rotations.

In this study it was also highlighted that pure sail pole-sitter orbits were not found, at least for the considered range of sail lightness numbers and other sail parameters. In fact, while orbits are sought minimising the propellant consumption, no path was found in which no propellant was needed.

3.6. Molniya orbits

The Molniya orbit is a type of highly elliptical orbit, with a period, typically, of one half of a sidereal day. Characteristic of the Molniya orbit is the fixed 63.4° or 116.6° inclination. At either of these critical inclinations the argument of perigee no longer rotates due to Earth's oblateness, and the position of apogee remains unchanged[7]. Molniya orbits are natural and do not require any additional acceleration apart from that necessary for station-keeping and counterbalancing additional perturbations (Sun, Moon, atmospheric drag), and therefore the mission lifetime can be long.

Although one spacecraft in this orbit does not provide continuous coverage of the pole, these orbits have been historically used for high-latitude telecommunications and observation. Moreover, a constellation of at least three spacecraft is sufficient to guarantee the continuous coverage.

A reference 63.4° Molniya with a period of 12 h is considered for this analysis [8]. The orbit is plotted in an Earth-centred frame in Fig. 9 (dashed line). The latitude of three spacecraft (phased 120°) on this orbit and the latitude Λ are represented in the dashed lines in Fig. 10: continuous visibility is available above 65°.

The short period of these orbits make them not suitable for static polar observation. In addition, a continuous telecommunication link would require steering of the ground station antenna to follow the spacecraft, as well as switching from one spacecraft to the next in the constellation at least three times for each orbital period. The period can be increased, at the cost of a higher distance of the spacecraft.

For the reference Molniya, the distance of the spacecraft when in sight with the North Pole is in the order of 30,000 to 40,000 km, comparable to the distance of the GEO.

Furthermore, due to the low altitude of the pericentre of the orbit, and consequently orbital angular velocity, one constellation would provide suitable coverage of one pole only.

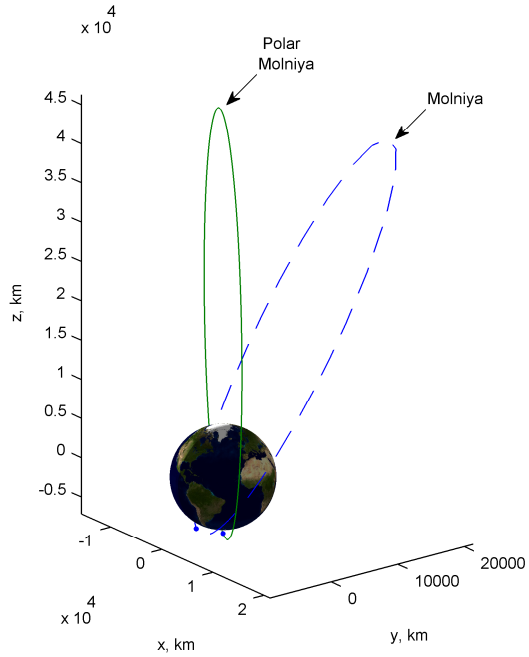

Fig. 9. Standard Molniya (63.4°, dashed line) and polar Molniya (solid line) orbits.

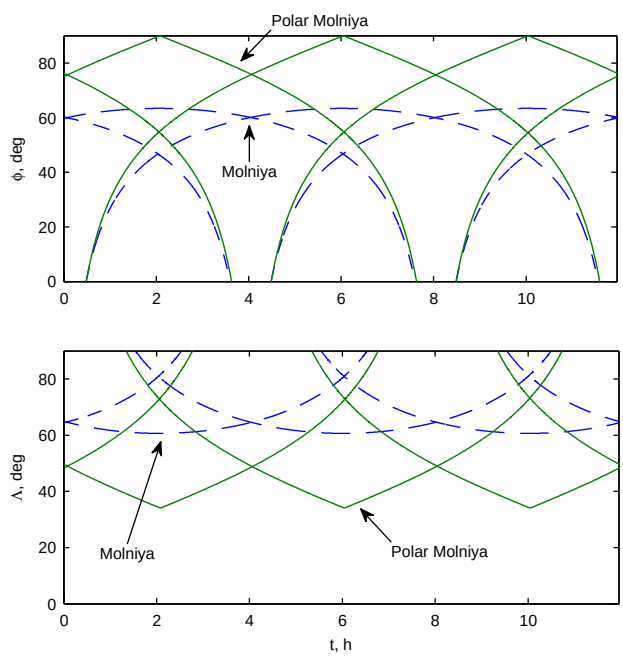

Fig. 10. Latitude and minimum continuous visibility latitude Λ of three spacecraft (phased 120°) in standard Molniya (dashed lines) and polar Molniya (solid lines).

3.7. <u>Polar Molniya and Taranis constellation</u>

The concept of polar Molniya was recently introduced by Anderson and Macdonald [8, 18]. The novel idea is to use continuous low-thrust propulsion to artificially change the critical inclination of the Molniya orbit. To increase polar

observation capabilities, the straightforward choice is to select an inclination of 90° (polar orbit). At this inclination, it is found that an acceleration of 0.0804 mm/s^2 is required, corresponding to 80.4 mN for a 1000 kg spacecraft.
The acceleration is unlikely to be provided by a solar sail: firstly, the short period of the orbits would require very fast slew manoeuvres; secondly, the sail has constraints on the magnitude and direction of the force that can provide; finally, at pericentre the aerodynamic drag might have a non-negligible effect on the large sail area, causing decay of the apocentre.
The authors foresee a spacecraft weighing about 2500 kg with a lifetime slightly longer than 5 years, carrying a payload of 1000 kg. These figures are similar to the data that can be extracted from the mass budget in Fig. 4, which can also be used to access different lifetimes or payload size for this concept. Note that due to the constant acceleration required, the lifetime is upper-limited to 9.5 years with the adopted technology, regardless the initial mass or payload mass.
The range of distances is very similar to that found for the classic Molniya orbit.
Anderson and Macdonald also defined a constellation of three spacecraft in a 12-hour polar Molniya orbit (see continuous line in Fig. 9), and they named it "Taranis". The visibility offered by the Taranis constellation is illustrated with continuous lines in Fig. 10, and compared to the standard Molniya: the maximum value of Λ (worst case) decreases to 49°, therefore overlapping to the latitudes covered by the GEO platforms (55°).

3.8. Vertical eight-shaped orbits

Vertical or eight-shaped orbits are families of periodic orbits that are connected to the L_1 and L_2 Lagrangian points of the circular restricted three-body problem. Two orbits are represented as D in Fig. 2. They are symmetric with respect to both the *x-y* and the *x-z* plane in the synodic reference frame, which means that they maintain the same orientation with respect to the Sun-Earth relative position. They have twice the multiplicity of conventional halo orbits, i.e. they cross the *x-z* plane four times within one period, hence the figure of eight.
These orbits can be found by continuation over one parameter: starting from a periodic orbit, one of its initial conditions is changed slightly, and a differential corrector is then employed to find the other initial conditions such that the resulting orbit is periodic. The procedure is repeated until a whole family of orbits is found.
These orbits were first discovered in 1920 by Moulton, and later found during a systematic numerical investigation of periodic orbits in the CR3BP started in the late seventies by Kazantzis [15, 19] for the Sun-Jupiter system, where they are classified as "Type C" orbits.
The use of these orbits as vantage points in the Sun-Earth system was first proposed by Folta et al. [20] in 2001, and in the Earth-Moon system by Archambeau et al. [21] together with their manifolds for lunar observation.
Recent work by the authors focused on the study of these orbits for their exploitation for quasi-static polar observation.
Each orbit in the L_1 or L_2 family can be characterised by its amplitude in z (z_0). The reason for which these orbits are considered for continuous polar observation is that, for small amplitudes, they start oscillating vertically at L_1 and L_2, but as their amplitude increases, they bend towards the Earth (see Fig. 11). Since the velocity of the spacecraft is lowest at the top and at the bottom of the orbit, it results that a relatively large fraction of the period is spent above the north and south poles. The period also increases with amplitude (solid line in Fig. 12).

It is to be noted that the L_1 and L_2 families are almost symmetric with respect to the Earth for the amplitudes of interest, therefore the result we present refer both to the L_1 and L_2 points, with only negligible differences.

Orbits with larger amplitude offer better visibility of the polar caps, due to increased bending towards the Earth, and due to their longer period. However, high amplitudes also mean higher observation distances, and the transfer to the higher amplitude orbits might also be more expensive.

Analogously to halo orbits, eight-shaped orbits also possess manifolds, i.e. surfaces of trajectories that can be obtained by integrating forward or backward in time initial conditions obtained by slightly perturbing a state on the orbit in the direction of the eigenvector associated to the stable or unstable eigenvalue. Unstable manifolds propagated backwards in time are made of trajectories that naturally wind onto the periodic orbit, and only an infinitesimal impulse is needed to finalise the transfer. Therefore, if one or more of these trajectories pass close to the Earth, they offer a valuable opportunity for transfer: only a single impulse in the vicinity of the Earth is necessary to inject the spacecraft into the manifold (which can be provided by the launcher or an upper stage), and the rest of the transfer is ballistic. This procedure has been widely used in the literature to compute low-cost trajectories to halo orbits [22].

In the same fashion, manifolds towards the Earth were computed for all the orbits in the family. The solid line in Fig. 13 represents the minimum distance from the Earth that is reachable from any of the manifolds of an orbit, as a function of the orbit amplitude. The curve has a deep minimum at about z_0 = 1.5 million km, and the manifolds of this orbit pass extremely close to the Earth's surface. Therefore this specific orbit has the important property that a low-cost transfer possible, with essentially only one impulse from a LEO orbit. Transfer to other amplitude orbits are certainly still possible, however the impulse needed at the end of the transfer is not negligible. It was found that the orbit at L_2 offers the same transfer possibility at approximately the same amplitude.

The orbits are purely ballistic, and therefore do not require any active propulsion. However, a preliminary analysis of the Floquet multipliers revealed that these orbits are unstable, therefore an accurate investigation of the amount of acceleration necessary for station-keeping will be required.

As for the Taranis constellation, to guarantee continuous view of the poles of the Earth, at least three spacecraft are necessary (spaced 120°). The advantage with respect to the Molniya orbits is that in this case the same visibility features are available for both the north and south poles. However, for amplitudes of about 1.5 million km, in which low-energy transfers are feasible, the visibility performances of these orbits are poor when considering the minimum elevation angle of 27°.

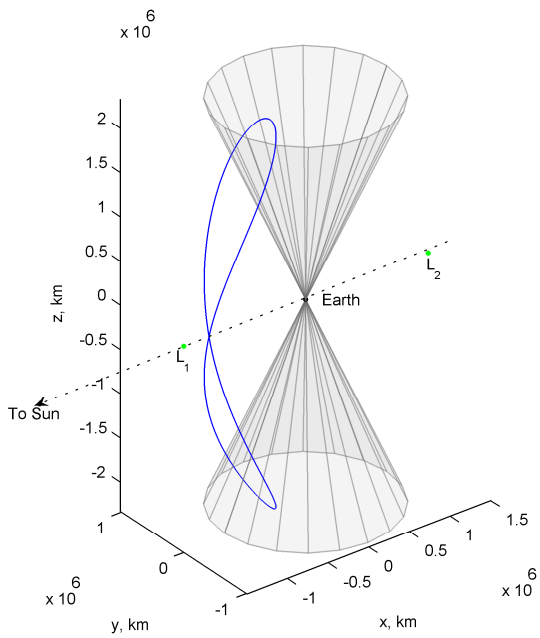

Fig. 11. An eight-shaped orbit at L_1. Note that the orbit intersects the polar axis cone in four points, while an ideal pole-sitter would stay on its surface.

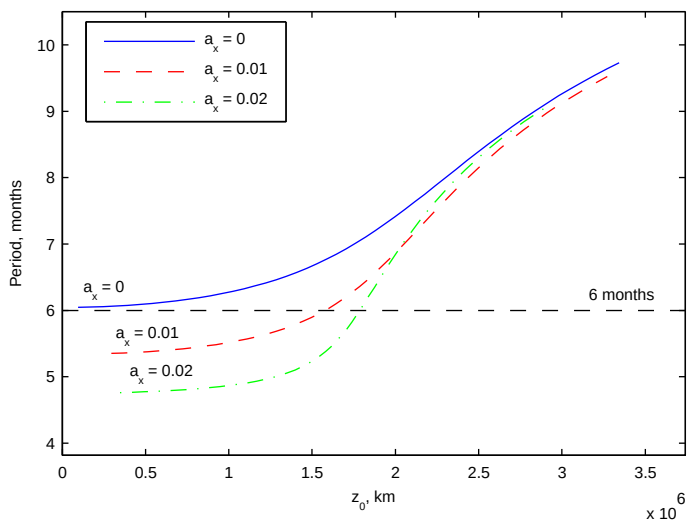

Fig. 12. Period of natural (solid blue line) and displaced (dashed red and green) eight-shaped orbits, as function of the *z* amplitude.

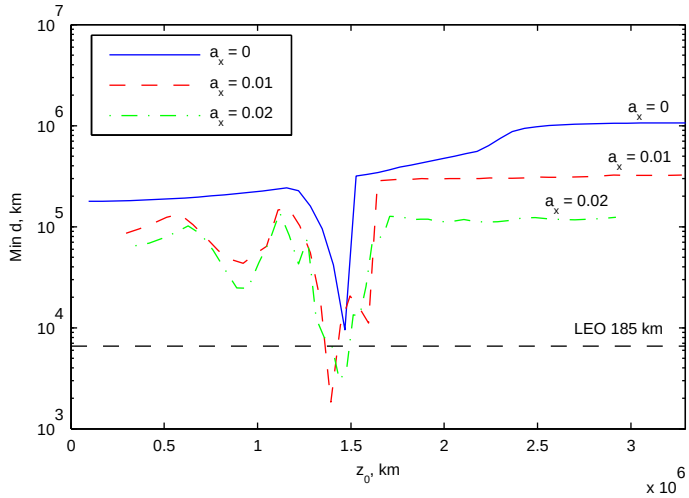

Fig. 13. Minimum distance from the Earth centre that is reachable from any of the manifolds of an orbit, as a function of the orbit amplitude. The black dashed line represents the distance of a LEO at 185 km of altitude.

3.9. <u>Displaced vertical eight-shaped</u>

Visibility performances can be increased by artificially displacing the orbits presented in the previous section, in a similar fashion as what was done by Waters [5], creating orbits similar to the one identified with E in Fig. 2.

It is well known that Lagrangian points can be artificially displaced by adding an acceleration term to the equations of motion. If a uniform and constant acceleration along the x axis (Sun-Earth line direction) is added, the Lagrangian points L_1 and L_2 are displaced as shown in Fig. 14.

Noting that a solar sail facing the Sun, in the vicinity of the Earth, produces an acceleration that is approximately constant in magnitude (since the distance from the Sun is approximately 1 AU) and directed towards the positive x axis (as long as the displacements in y and z are small), we can state that the constant acceleration in x is very similar to the effect of applying a Sun-facing solar sail to the spacecraft.

It follows that orbits around L_2 would be displaced towards the Earth, which is a desirable property. On the other hand, orbits at L_1 would displace further from the Earth, limiting the choice of the Lagrangian point to L_2.

In Fig. 14 and the following, the acceleration a_x is expressed in non-dimensional units, because in the Sun-Earth system at about 1 AU, its value is approximately equal to the lightness number β of an ideal solar sail generating it.

The orbits in the previous section were recomputed, including the additional acceleration term of 0.01 and 0.02. As the sail is flat towards the Sun, the dimensional acceleration is also the characteristic acceleration of the sail: it can be computed by using Fig. 1, and results in 0.06 and 0.12 mm/s^2 respectively. It is found that, despite the displacement of the orbit in the vicinity of L_2 being modest, a higher displacement is obtained at the top and the bottom of the orbit, as it appears to bend more towards the Earth (see Fig. 15, where the family of natural orbits is compared with the two families of the displaced orbits).

It is found, for example, that for z_0 = 1.5 million km, while the natural orbit does not intersect the cone described by the polar axis, an orbit with the same amplitude but displaced with a_x = 0.02 has its apex above the Earth, with

considerable visibility improvement. At any fixed amplitude, the period also changes due to the additional acceleration (see again Fig. 12).

It was also found that almost-natural transfers to the Earth are possible even for displaced orbits, at roughly the same value of the amplitude (around 1.5 million km). Manifolds fall onto the Earth naturally when no acceleration is applied along the manifold itself, meaning that the spacecraft would deploy the solar sail only once arrived to the eight-shaped orbit (see dashed lines in Fig. 13).

An interesting effect of the artificial displacement is that it is now possible to find orbits whose period is exactly 6 months (see dashed lines in Fig. 12): this means that the spacecraft is synchronous with the apparent precession of the polar axis, and therefore it has the same viewing conditions on the Earth every year.

Visibility of the poles for the three-spacecraft constellation is illustrated in Fig. 16, in which the two values of acceleration are respectively considered. In each, the orbit with a period of 6 months was considered as a reference. The 1-year period oscillation is due to the apparent rotation of the Earth's polar axis in the synodic frame. For a_x = 0.01, the coverage is barely sufficient to continuously cover the pole itself, and therefore the minimum value of Λ approximates 90°. However, visibility conditions improve substantially with a_x = 0.02: three spacecraft on the 6-month orbit can continuously cover all latitudes above 65° (both north and south), with elevation angle of at least 27°. Alternatively, by using a slightly wider orbit, it is possible to lower the latitude to 55°, thus overlapping with the standard coverage band around the equator provided by GEO services. However, in this latter case, the coverage pattern would change yearly due to the longer period of the orbit.

As before, the mass of a potential spacecraft in one of these orbits can be computed considering the sail mass budget in Fig. 7, remembering that $a_x \approx \beta$. It is interesting to note that the lightness numbers considered are much smaller than those proposed in previous approaches with sails, therefore enabling the use of lower-performance sails. We also believe that slightly higher values of β, around 0.05, would enable to displace these orbits even further, extending the coverage latitude. However, these orbits have not been computed yet.

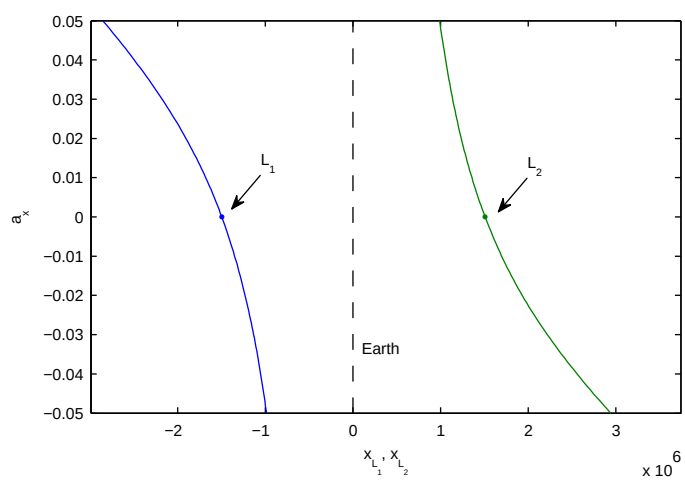

Fig. 14. Displacement of L_1 and L_2 points under a uniform and constant acceleration in the *x* direction.

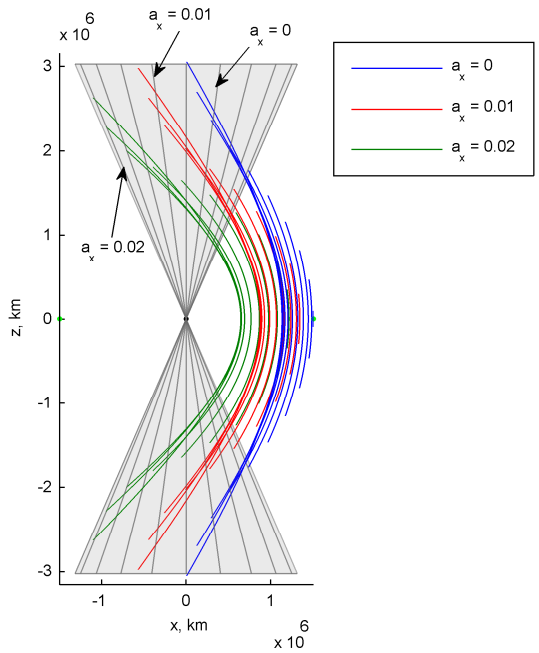

Fig. 15. Natural and displaced families of eight-shaped orbits around L_2.

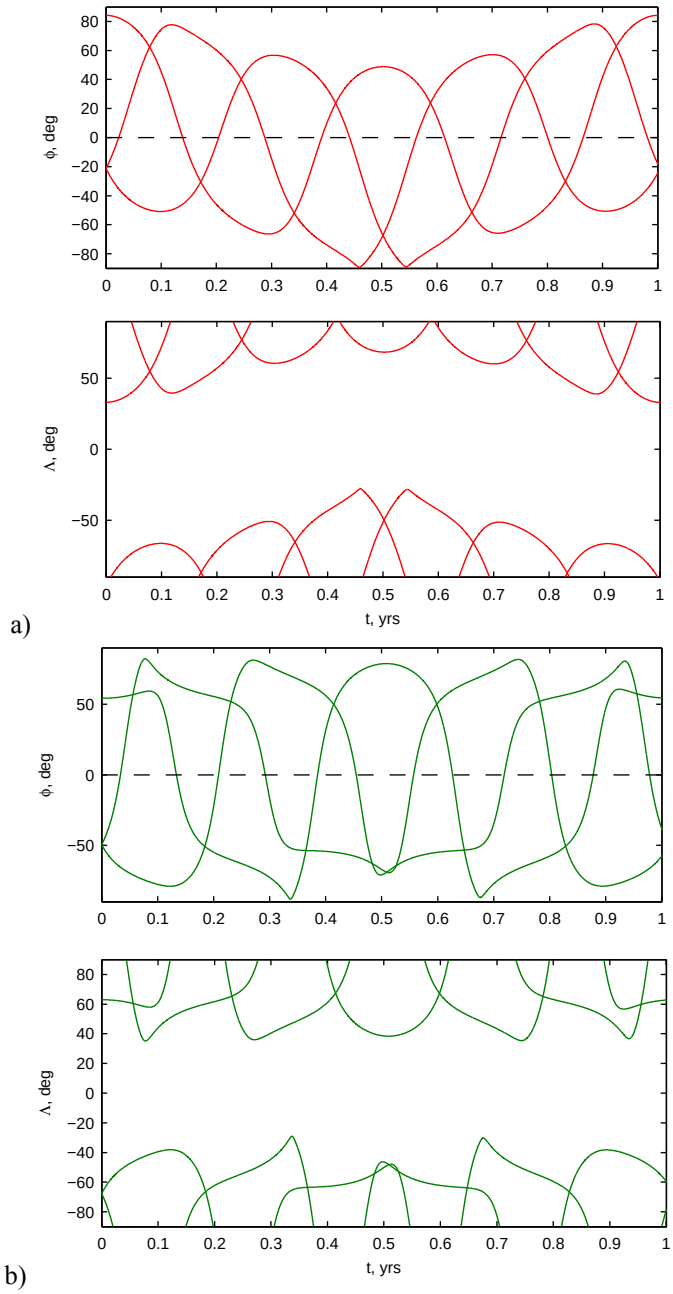

Fig. 16. Spacecraft latitude (φ) and minimum continuous visibility latitude (Λ) of three spacecraft (phased 120°) in displaced eight-shaped orbits with period 6 months. (a) refers to acceleration a_x = 0.01; (b) a_x = 0.02. Note that a low-energy transfer is possible to the orbit in (a).

3.10. Summary

Table 1 attempts to summarise the main features of each concept presented in this paper. The columns represent respectively: the range of distance from the Earth, the lifetime and the initial mass for a 100 kg payload, the period of the orbit (if any), the type of propulsion system employed, the average SEP acceleration, the number of spacecraft necessary for coverage of either or both poles, and the maximum angle Λ (worst case) achieved during the mission.

It shall be underlined that the values presented here are indicative and referred to their own specific cases and scenarios, and therefore care shall be taken when comparing them directly. We refer to each paper describing the work for a detailed description and accurate data.

For spacecraft with an SEP system, the specific impulse is 3000-3200 s. For spacecraft with a sail, σ_s = 10 g/m^2 except for the hybrid pole-sitter, in which 7.5 g/m^2 is considered

The mass budget, for sake of comparison, was computed according to Fig. 4 for SEP spacecraft, Fig. 7 for sail spacecraft, and data in Ceriotti [16] and Baig [6] for hybrid spacecraft. Note therefore that the mass for the SEP pole-sitter differs from what was presented by Driver [2] in his work.

Note the substantial saving in initial mass for all the sail-only systems, compared to systems carrying SEP. However, the former have substantially lower technology readiness level: these results should therefore push the research and development of such promising systems. In addition, the lifetime of a sailcraft is in principle not limited by propellant mass, and it can be substantially longer than that of the SEP systems. However, it is envisaged that even sailcraft could require propellant for attitude control and station-keeping, therefore limiting their lifetime. The same applies for spacecraft in Molniya and eight-shaped orbits, that do not require any acceleration in the nominal case, but would definitely require thrust for station-keeping. A 10% fraction of the payload mass was added for these systems.

Also note the higher mass value of the hybrid pole-sitter: this is due to the 8-year lifetime that was considered: as clarified in Ceriotti et al. [16], the hybrid propulsion allows mass saving when longer missions are designed. The same 8-year mission with a pure SEP spacecraft would require 3773 kg.

Table 1. Summary of the mission concepts and main features. Notes: * Thrust is needed for stationkeeping and counteract perturbations. ° For hybrid, average on first year. On following years SEP acceleration is lower due to mass decrease and hence increased contribution from the sail.

Concept and main author	Distance, km	Lifetime, yrs	Mass, kg (m_{pl} = 100 kg)	Period	Propulsion system	Avg.° SEP acc., mm/s^2	S/C 1 pole	S/C both poles	Max. Λ (ε_{min} = 27°)
SEP pole-sitter (Driver)	2.1×10^6	5	513.76	1 y	SEP	0.17	1	2	27°
Statite (Forward)	2.6×10^6	/	148 (σ_s = 10 g/m^2)	/	Sail* (β = 0.05)	/	/	/	/
Orbit around displaced equilibria (Waters)	1.5-2.6×10^6	/	150 (σ_s = 10 g/m^2)	1 y	Sail* (β = 0.051)	/	1	2	88°
Artificial hybrid equilibria (Baig)	2.7×10^6	5	288 (σ_s = 10 g/m^2)	/	Hybrid sail/SEP (β = 0.03)	0.09	2	4	60°
Optimal hybrid pole-sitter (Ceriotti)	1.9-2.7×10^6	8	2871 (σ_s = 7.5 g/m^2)	1 y	Hybrid sail/SEP (β = 0.025)	0.13	1	2	27°
Molniya	36,000	/	110 kg	12 h	None*	/	3	6	65°
Polar Molniya/Taranis (Anderson)	36,000	5	180	12 h	SEP	0.08	3	6	49°
Eight-shaped orbit (natural transfer)	1.4-1.8×10^6	/	110	6.5 months	None*	/	3	3	/
Displaced eight-shaped orbit (Ceriotti)	0.8-1.8×10^6	/	115 (σ_s = 10 g/m^2)	6 months	Sail* (β = 0.02)	/	3	3	65°

4. FUTURE APPLICATIONS

Potential future applications of the concepts described above are briefly discussed here. We mainly see applications which exploit the platform for polar Earth observation or for high latitude telecommunications.

4.1. Earth observation

The utility of polar orbits and polar stationary spacecraft depends largely on the view that can be obtained. In particular, while the view of the Earth from GEO distance is well known, less assessment has been made of the quality and resolution of the images that can be taken from 1 million km or further. The Deep Space Climate Observatory (DSCOVR) spacecraft (formerly Triana) was designed with a 30.5 cm aperture near-IR to ultraviolet imager with an expected resolution of 8 km for some data products [23].

4.1.1. Resolution

Fig. 17 shows the maximum resolution achievable (limited by diffraction [7], and not taking into account pointing stability of the camera) on the ground from a range of distances, from 30,000 km (about the apogee of the Molniya and polar Molniya orbits) to 3 million km (farthest point of pole-sitters and static concepts). Clearly, the resolution that can be provided by these two groups of platforms is very different. For the Molniya, resolution of 100 m to 1 km is easily achieved with a small instrument in the visible and IR bands. When considering spacecraft at millions of km from Earth, resolution is limited; however, in the visible range, resolutions around 20 km should be readily available, even considering the platforms at 3 million km. Resolution less than 1 km is very unlikely for pole-sitters, and less than 500 m for the spacecraft in 8-shaped orbits. In the infrared range, the resolution is degraded, leading to resolutions in the range of 10-70 km at 3 million km from Earth, depending on the instrument size. In the shortest wavelength in the microwave band (λ = 300 μm), resolution is an issue. To obtain a resolution of less than 100 km (values higher than this are not expected to be interesting) at million km, then an instrument of at least 10 m is necessary. Note that while this size is completely unrealistic for an optical lens, it is not unreasonable in the microwave range. It can be envisaged, for example, to integrate a large-diameter antenna into the solar sail.

To give an idea of what can be seen from a distance of order of magnitude of million of km, Fig. 18 shows four different pictures of the Earth, taken by the spacecraft Galileo. The resolution of large-scale weather systems from such distances is surprisingly good. Note that the purpose of the spacecraft was not Earth imagery, and these pictures were taken during an Earth swing-by. Therefore we imagine that higher resolution can be easily achieved by using an instrument that is calibrated for the Earth and with the right focal length.

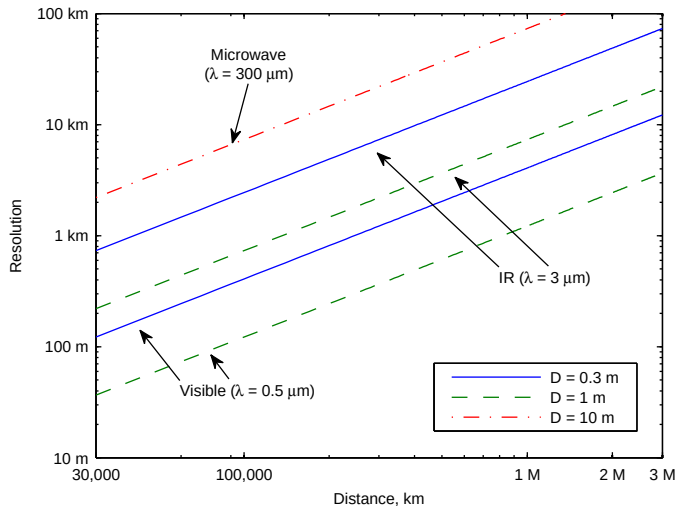

Fig. 17. Diffraction-limited resolution as a function of spacecraft distance, for different wavelengths (λ) and instrument aperture diameter (D).

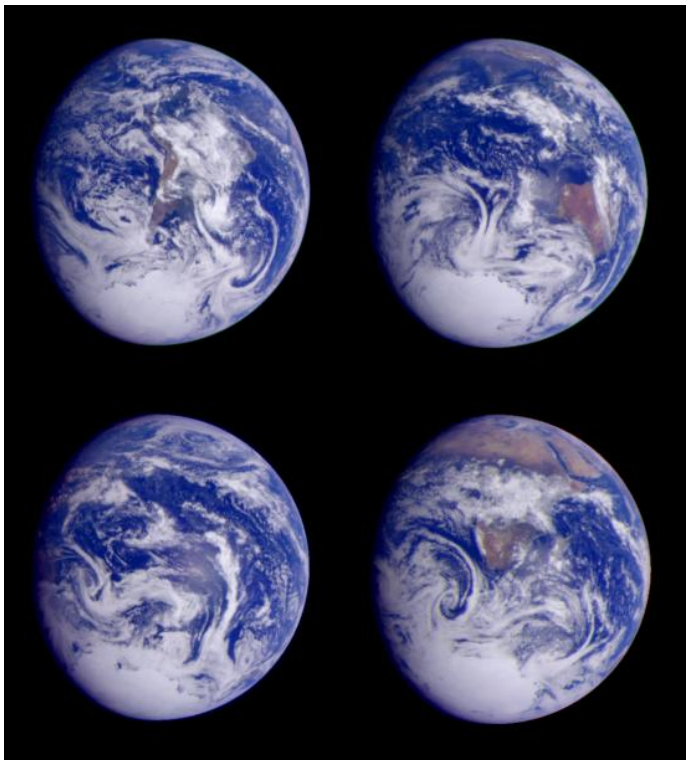

Fig. 18. Four views of the Earth as taken by Solid-State Imaging instrument on Galileo spacecraft during its first Earth fly-by, at six-hour intervals on December 11, 1990, at a range of between 2 and 2.7 million kilometres. Each disc is about 500×500 pixels. (Credit: NASA, Johns Hopkins University, http://photojournal.jpl.nasa.gov/catalog/PIA00728)

4.1.2. Potential applications

The most interesting application is polar weather forecasting. As outlined very clearly by Lazzara et al. [24], the benefits of having a continuous and hemispheric view of the pole are enormous. First of all, we would be able to obtain continuous views of dynamic phenomena and large-scale polar weather systems. Recently GOES-13

geostationary spacecraft captured an image of the Earth allowing a clear view of hurricane Irene (Fig. 19). Similar images, in lower resolution, could be obtained for the poles from stationary platforms or highly eccentric orbits, but not from low and medium orbits, due to the lengthy periods of time necessary to reconstruct the whole image from the swaths.

Composite images are nowadays also used to create so-called atmospheric motion vectors for the identification of storm systems [24]. However, composites introduce gap problems related with geolocation and intercalibration [1]. An image of the pole, even at a spatial resolution of 10-25 km is therefore very desirable.

Glaciology and ice pack monitoring would also benefit from polar observation. McInnes et al. [1] envisage that, by using a multispectral infrared sensor and with a spatial resolution of order 15-20 km, applications for ice mass studies are also possible, observing the dynamical linkages between air masses and water vapour/cloud advection, and the moisture transport over the high Antarctic plateau.

The International Polar Year (IPY) was an ambitious science programme involving more than 60 countries, carried out in 2007 and 2008. By carefully piecing together billions of radar data points that were collected over Antarctica by satellites, including Envisat (ESA), Radarsat (CSA) and ALOS (JAXA), a team of scientists has created the first map of ice motion over the entire continent of Antarctica[2] (see Fig. 20). It is possible that a continuous hemispheric view, even at low resolution could consistently improve the quality and amount of information obtained for monitoring the ice sheet.

Ultraviolet imagery of the polar night regions at 100-km resolution or better would enable real-time monitoring of hot spots in the aurora that can affect high frequency communications and radar. Since auroral conditions can change rapidly, this could dramatically improve the current data, which is a composite of imagery and charged particle detector readings from polar low Earth orbit satellites that is taken over a few hours [24].

Lastly, polar spacecraft can be used during the polar night at infrared wavelengths, with limitations in resolution as described in Fig. 17.

[2] http://www.esa.int/esaEO/SEMCFKOT9RG_index_0.html [cited 12/09/2011].

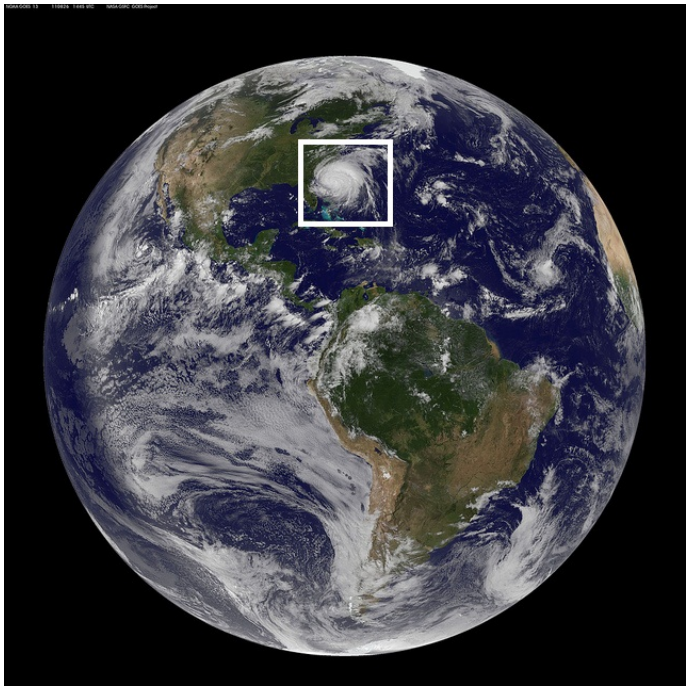

Fig. 19. Hurricane Irene (in the square) in an image taken by GOES-13 satellite on August 26, 2011 (Credit: NOAA/NASA GOES Project).

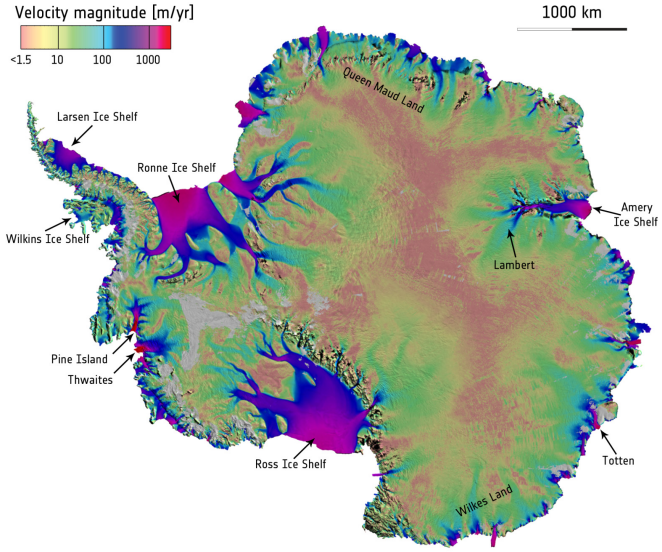

Fig. 20. Antarctic ice sheet velocity (credit: NASA Goddard Space Flight Center Scientific Visualization Studio).

4.2. <u>Telecommunications</u>

GEO platforms are currently used for high-bandwidth telecommunications, and therefore it is expected that similar performances in terms of data rate can be obtained by spacecraft in Molniya and polar Molniya orbits (while in sight with a ground station). However, it is expected that high bandwidth will not be available for spacecraft at distances

of the order of million km. This section will present a preliminary link budget and then introduce some potential applications.

4.2.1. Link budget

A preliminary link budget [7] was computed, to assess the size of the antenna and the power needed for telecommunications. Considering the data downlink scenario, at an operating frequency of 2.2 GHz, a 5.3-m ground station antenna, we can compute the approximate power needed for a given required data rate, depending on the antenna diameter (Fig. 21). Other assumptions used in this link budget are taken from the example in Wertz et al. Also, a pointing error of 1° is taken into account, so even for telecommunications, the accurate pointing of the spacecraft is important. Note that these values are provided as a rough estimation, and could be improved by designing accurately the telecommunication subsystem, both on the ground segment and in the space segment.
As foreseen, for realistic sizes of the on-board antenna (up to 2 m in diameter), we can have a power of about 1 kW for a data rate of 1 Mbps at 3 million km. However, a 1-m antenna needs about 200 W at 800,000 km, which is achievable with the eight-shaped orbits, but not with pole-sitters. A data rate of only 10 kbps results in more affordable values of power needed for transmission, even at the highest considered distance from the Earth (below 30 W for a 1-m antenna).

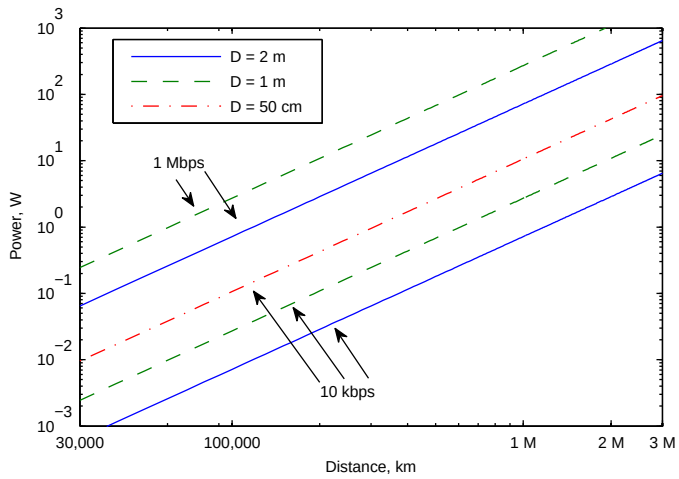

Fig. 21. Antenna power required as a function of distance, for different spacecraft antenna diameters (*D*) and data rates.

4.2.2. Potential applications

A number of potential telecommunications applications were proposed by McInnes and Mulligan [1] during a study in 2003. The first one consists on the use of polar stationary spacecraft as data relay for future NOAA polar orbiting satellites, such as the NPOESS system. They envisage the use of two Statite-like sailcraft, one for each hemisphere, two high-latitude ground stations (above 70° latitude) and three NPOESS spacecraft in Sun-synchronous near-polar orbits, generating at most 15 kbps of environmental data. In this way, the NPOESS satellite can provide continuous coverage using only two data relay spacecraft and two ground stations. The original plan for NPOESS employed 15 ground stations with downlink rates of 150 kbps because of the limited station visibility. If the relay spacecraft are

put on the day-side of the Earth, then a small gap only occurs for one NPOESS satellite at summer and winter solstice. Instead, for the night-side, continuous coverage is not possible due to the worst displacement that can achieved with the same lightness number (see again Fig. 5a). Further studies also concluded that cross-link communication between NPOESS and relay spacecraft appears technically feasible, with some modest enhancements to the NPOESS baseline design, even if they are not common on current weather satellites.

A second application that is envisaged is telecommunications with the polar regions. A polar stationary satellite system could significantly improve them, and this might become particularly useful in the southern hemisphere, where there are Antarctic research activities ongoing and the communication capabilities are limited. It is proposed 24/7 high-bandwidth (up to 45 Mbps) inter- and intra-continental link at locations where large ground stations are available, and a continuous 1-Mbps link at sites with smaller antennas. Possible utilisations include data links for scientific experiments, automated weather stations, emergency airfields, telemedicine, and real-time monitoring. The same study suggests the use of polar stationary spacecraft as a data relay for the Geostorm spacecraft, which would be positioned at about 0.95 AU from the Sun on the Sun-Earth line.

Finally, it is recent news that, for the second time since satellite measurements began in 1970, the Northwest Passage and the Northern Sea Route through the North Pole are open simultaneously[3]. As sea ice melts, we could foresee new shipping routes through the Arctic ice cap, and pole-sitters could also be used for ship tracking and telecommunications, and to support future high latitude oil and gas exploration.

5. CONCLUSIONS

The paper presented a survey of missions for continuous and hemispheric polar observation and telecommunications. It included mission concepts that exploit the two- or three-body problem, different propulsion systems (solar electric propulsion, solar sailing, and a hybridisation of the two), and different orbits (artificial equilibria, non-Keplerian orbits, Molniya and polar Molniya).

A novel way to displace and exploit eight-shaped high-amplitude vertical orbits at L_2, currently under investigation, was also presented. These orbits offer continuous monitoring of both the poles, by using three spacecraft with a solar sail. Even if the spacecraft are not stationary above the poles, they have a long dwell time at high latitudes (several months). However, the instability of the orbits might require an accurate control.

For all these concepts but the (polar) Molniya, the main drawback is the relatively large distance of the spacecraft from the Earth. This imposes requirements and limitations for the payload, for example in terms of pointing accuracy, imaging resolution and available bandwidth. However, similar issues arise with imagers proposed for the classical L_1 point such as the Deep Space Climate Observatory (DSCOVR) mission.

Stationary spacecraft, like pole-sitters, require to be in the rage of million km from the Earth (for reasonable required acceleration and lifetime), but offer a truly continuous view; other concepts, like polar Molniya orbits, use spacecraft that are much closer to the pole, however they move relatively fast with respect to the Earth. The eight-shaped orbits are a compromise, in the sense that the spacecraft does oscillate between north and south poles, but stays almost-stationary for a long time (order of months) above each pole.

[3] http://www.esa.int/esaEO/SEMT7TRTJRG_index_0.html [cited 12/09/2011].

Possible applications were identified in medium-resolution polar Earth observation (mainly for weather forecast and glaciology) and telecommunications (data relay of other spacecraft, bases in Antarctica or future high latitude shipping routes, and oil and gas exploration).

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